



Sentinels

This LWS program element includes the Inner Heliospheric Sentinels (IHS) and the Far Side Sentinel (FSS) missions, each with its own distinct concept.

*And post o'er land and ocean without rest,
They also serve who only stand and wait.*

*On His Blindness
John Milton*



Goddard Space Flight Center



Sentinels Study Chronology

Date	Event	Concept
1/13/00	Sentinels Briefing By A. Szabo	Far Side Observer, L1 Cluster, NG STEREO
1/20/00	Preformulation Team Meeting	Imaging/In-Situ Libration Point Measurements
2/2/00	Preformulation Team Meeting	Emphasis On Heliospheric Elements
2/3/00 to 3/31/00	Libration Point Orbit Studies	Orbit Trajectory Options
3/31/00	Preformulation Team Meeting	Far Side Sentinel/4 Inner Heliospheric Sentinels
4/6/00	Sentinels Workshop	4 Satellite Constellation/Far Side Options
4/7/00	JPL Preliminary FSS Concept	3-Axis Spacecraft/5 Instruments/1 Launch
4/10/00	IMDC IHS Briefing	4 Spinning Satellites/4 Instruments/1 Launch
4/17/00 to 4/20/00	IMDC IHS Study	4 Spinning Satellites/4 Instruments/1 Launch
4/20/00 to 4/27/00	JPL FSS Concept/Cost Update	Custom Spacecraft/5 Instruments/1 Launch
4/21/00 to 5/5/00	GSFC Mission Costing	Far Side Sentinel/4 Inner Heliospheric Sentinels
5/25/00	FSS Mission Concept Summary	Defined In Attached System/Subsystem Charts
5/31/00	Program Operating Plan	Costed As Two Separate Missions





Sentinels Concept Evolution

LWS scientists initially considered a complement of spacecraft for the Sentinels mission that included a spacecraft observing the far side of the Sun, a cluster of satellites at L1, and two next-generation STEREO spacecraft. Both remote images of the Sun and in-situ measurements would be taken. After much deliberation, the LWS science pre-definition team endorsed a four satellite inner heliospheric constellation with a suite of in-situ instruments and a solar far side observer with both remote and in-situ type instruments.

Although supportive of an L1 element to carry on the measurements of ACE, such an element was viewed as operational in nature and thus not an integral part of basic LWS research objectives.





Sentinels Concept Evolution

Concept definition and costing for the Far Side Sentinel (FSS) was assigned to the Jet Propulsion Laboratory because of their previous studies of a Solar Far Side Observer. A GSFC IMDC study was then performed to develop a concept for the four Inner Heliospheric Sentinels (IHS) that would also be used for costing.

The same technical and cost templates were used by both organizations to standardize reporting.





Inner Heliospheric Sentinels



Goddard Space Flight Center



IHS Mission Profile

Description: Continuous observations of the heliosphere in order to characterize the environment through which solar transients propagate as well as to track the full evolution of space weather disturbances to improve forecasting

Instruments: Four in-situ instruments per spacecraft including a magnetometer, a solar wind analyzer, an energetic particle detector, and a radio waves instrument

Spacecraft: Four identical spacecraft spinning at 20 RPM normal to the orbit plane, each with a propulsion system for final orbit spacing

Launch Date: December, 2008

Mission Life: 3 years on station with an optional 2-year extension of mission operations as resources allow

Orbit: Four elliptical heliocentric orbits at various distances (0.5 to 0.95 AU x 0.72 AU) from the Sun, all approximately in the plane of the ecliptic

Space Access: One launch from ETR on a Medium Class ELV

Key Technologies: Smaller instruments and enhancing technologies at the subsystem or component level





IHS Mission Timeline

The following serial time spans are assumed for mission planning:

- Extended pre-formulation and formulation phase for instrument, spacecraft, and ground system accommodation studies to match the available funding profile
- 12 months for conclusion of project formulation and definitization prior to approval
- 4 years from approval to launch readiness
- Launch in December, 2008
- 4.5 month transit to final orbits
- 3 years for baseline mission operations including transit period
- 2-year mission extension (option for evaluation)





IHS Mission Objectives

The IHS mission relies on a small constellation of spinning spacecraft in concentric elliptical orbits at various distances from the Sun to achieve mission objectives.

Each spacecraft carries a complement of *in-situ* instruments that will assist in the global characterization of the ambient structure of the inner heliosphere and will follow the evolution of large-scale features. Scientific knowledge gained from this mission will improve lead time and accuracy of space weather forecasts.

Data from the IHS mission will be supplemented by the FSS mission that is described later in this document.





IHS Instrument Complement

The baseline IHS instrument complement for each spacecraft consists of the four instruments listed below and their associated electronics:

- A three-axis magnetometer mounted on a 3-meter boom
- A solar wind analyzer on a 1-meter boom
- An energetic particle detector
- A radio waves experiment utilizing two 12-meter wire antennas

Instrument system parameters, shown in the table that follows, are based on direct heritage from the WIND, ACE, STEREO, and MESSENGER missions.





IHS Instrument Resource Accommodations

Type/Classification	Size	Mass	Power	Data Rate
	LWH or DH (cm)	(kg)	Avg/Peak (W)	Avg/Peak (kbps)
Magnetometer*	5x5x10	5	2	0.1
Solar Wind Analyzer*	10x10x30	4.5	5	1.5
Energetic Particle Detector	10x10x15	4	3.5	0.2
Radio Waves Instrument	20x20x20	5.4	5	1.0
Digital Processing Unit	25x25x20	10	10	
Total		28.9	25.5	2.8
* Includes mass for boom				





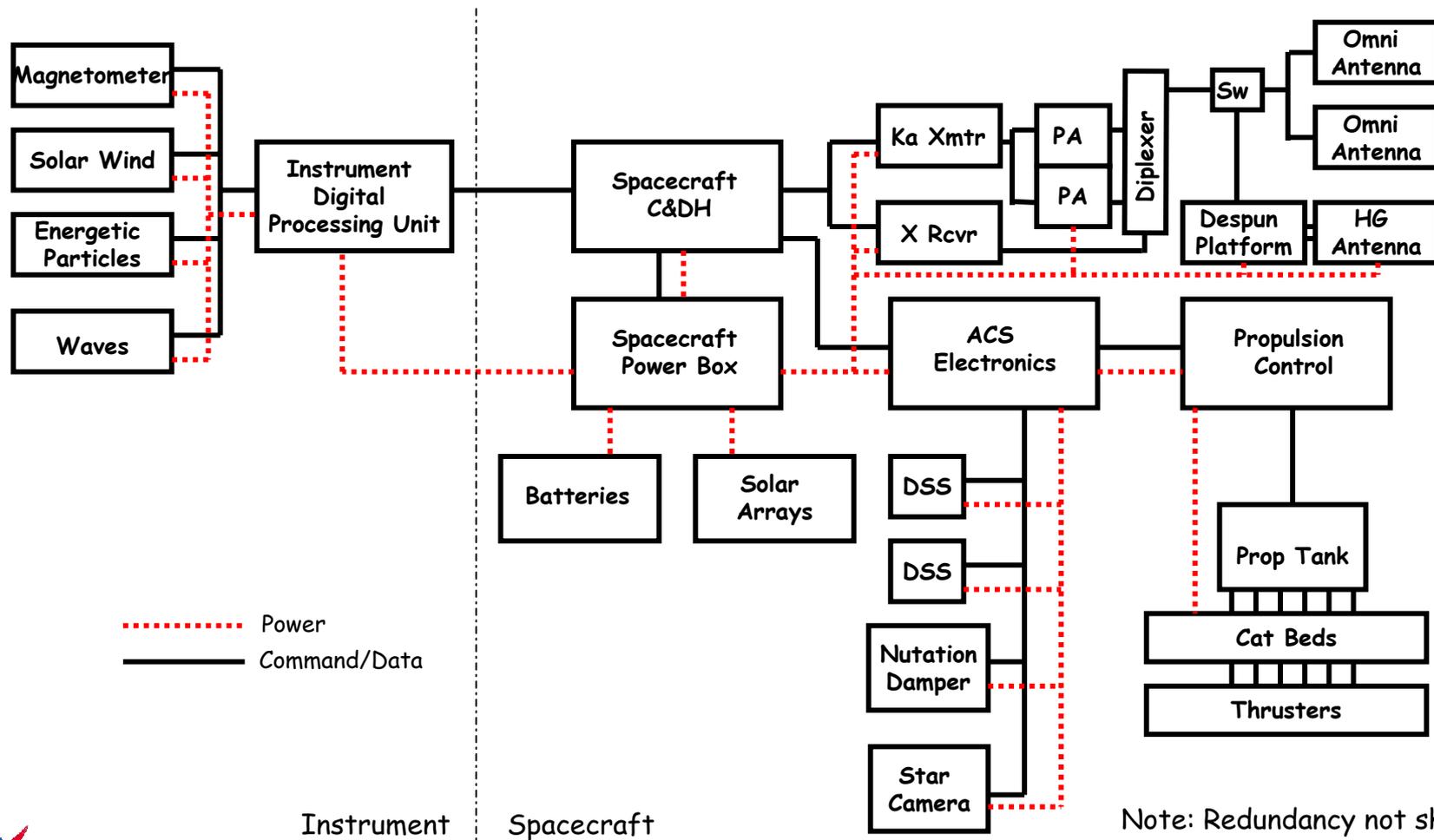
IHS System Synopsis

- Four spinning spacecraft with a complement of *in-situ* instruments are placed into heliocentric orbits at various distances from the Sun in order to characterize the three-dimensional nature of the heliosphere.
- The spacecraft are delivered to orbit by a single launch vehicle with the required performance for a C3 of $10 \text{ km}^2/\text{sec}^2$. There is a 40 day launch window available for this C3 value.
- A Venusian Gravity Assist (VGA) is employed to disperse the spacecraft orbits and an on-board propulsion system is used to provide the final orbit spacing.
- A three year mission design life is baselined with consumables allotted for two additional years.
- The DSN is used for regular downlink of science data with a 2 week on-board storage capacity for those times when spacecraft communication to Earth is precluded.





IHS System Block Diagram





IHS Mass Summary

Element				Mass (kg)
Instrument Payload				61
Baseline Instruments			29	
Wire Dispensers (2)			30	
Instrument Harness			2	
Spacecraft Bus				186
	Mechanical		75	
	Thermal		5	
	Power		27	
	Attitude Control		8	
	Propulsion		13	
	C&DH		8	
	Communications		22	
	Despun HGA		25	
	Harness		3	
Balance Mass				8
Propellant				10
Unit Spacecraft Mass				265

Values are best estimates and do not include contingency.





IHS Launch Mass Evaluation

The baseline mission concept is to launch all four IHS spacecraft on one launch vehicle. The total mission mass budget for this single launch scenario is given below.

Mass of Four Spacecraft	1060 kg
Satellite Dispenser	<u>86 kg</u>
Total Mass to Orbit	1146 kg
Delta II 7925H-10 Lift Capability for $C_3=10 \text{ km}^2/\text{sec}^2$	1170 kg
Launch Mass Margin	2 % (inadequate)

Mass margin must be increased to an acceptable level. Options for further study include potential reductions in instrument and spacecraft subsystem masses, deletion of the Radio Waves instruments, or, as a last resort, elimination of one spacecraft. As a risk mitigation strategy, the additional cost of the next larger launch vehicle was included in the resource analysis pending further study. This larger launch vehicle (Delta IV) would also allow inclusion of the FSS spacecraft on the same launch if programmatically feasible.





IHS Power Summary

Element				Power (W)
Baseline Instruments				26
Mission Unique Spacecraft Bus				88
	Thermal		0	
	Power		15	
	Attitude Control		6	
	Propulsion		1	
	C&DH		20	
	Communications		41	
	Harness		5	
Total				114
Solar Array Capability at 0.95 AU (BOL)				288
Power Margin (BOL)				153%
Solar Array Capability at 0.95 AU (EOL)				230
Power Margin (EOL)				102%

Values are best estimates and do not include contingency.

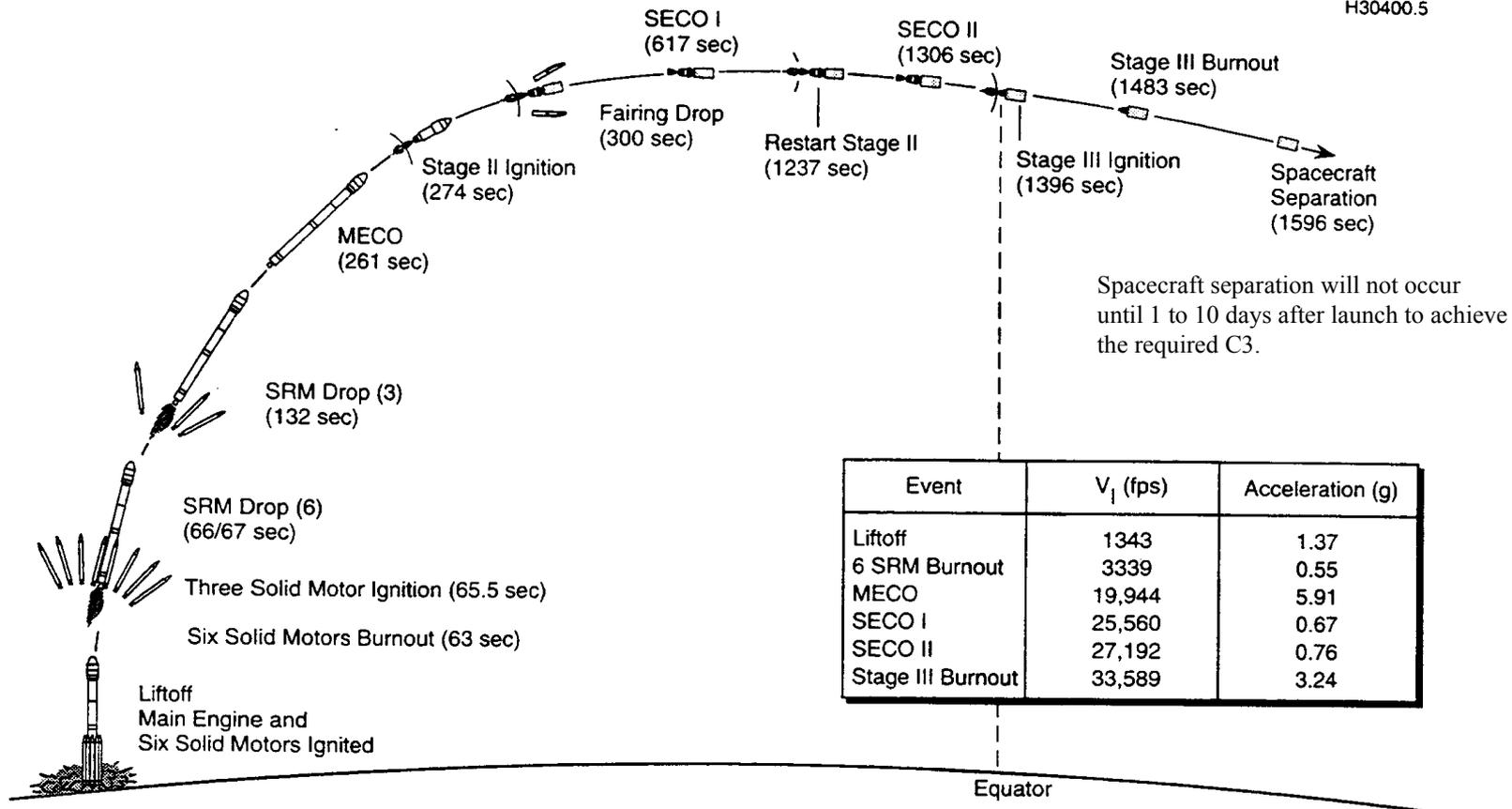
Note: No shadow periods occur for the mission orbits.





IHS Delta 7925 Launch Profile

H30400.5



Eastern Range launch site, flight azimuth 95 deg; maximum capability to 28.7-deg inclined GTO, 100-nmi perigee





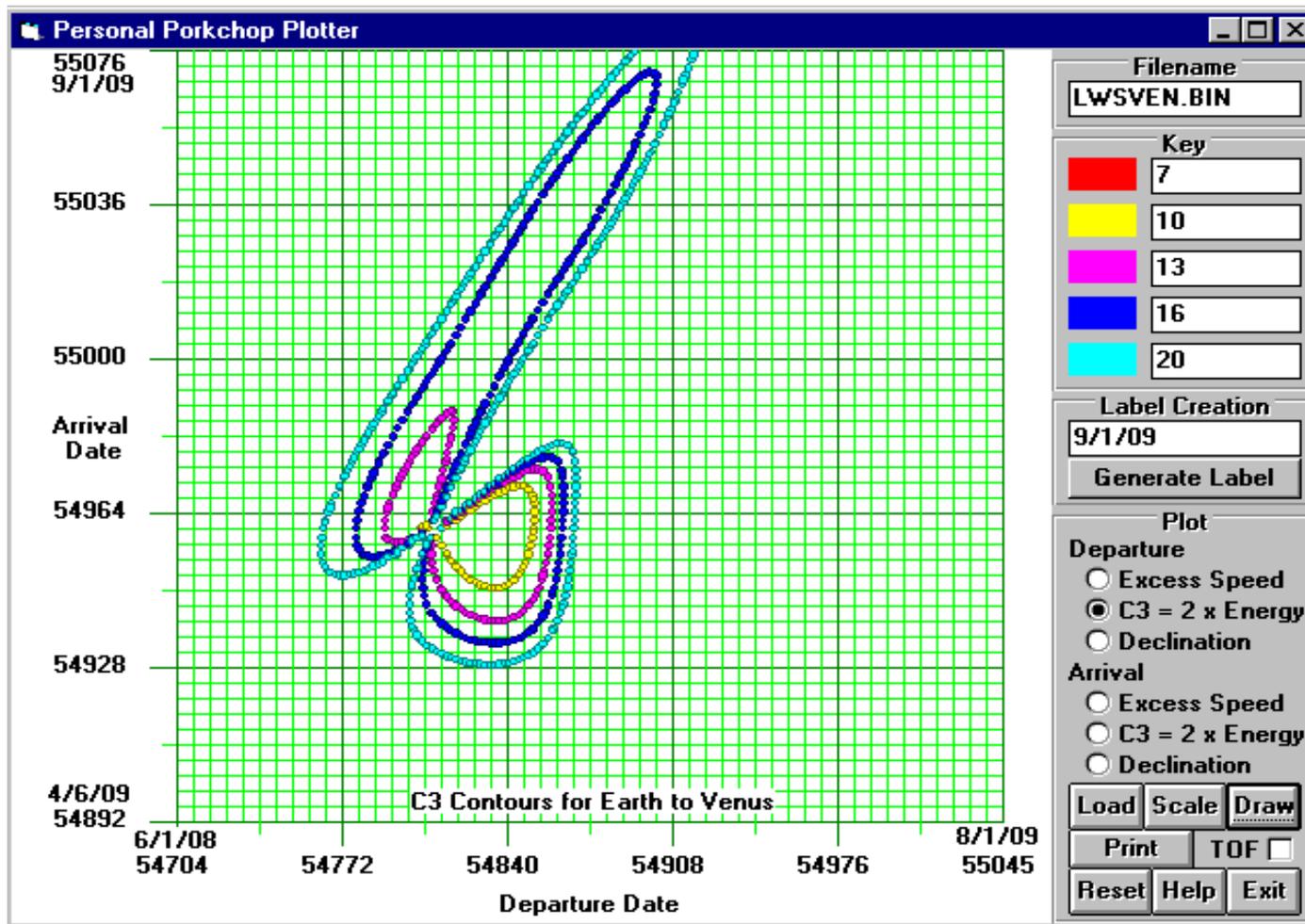
IHS Launch Trajectory And Orbit Parameters

- Minimum C3 of $7.33 \text{ km}^2/\text{sec}^2$ for launch on 12/25/08 with a C3 of $10 \text{ km}^2/\text{sec}^2$ for approximately 40 days thereafter
- Separation from launch vehicle and thruster firings at 1 to 10 days after launch to place spacecraft on correct trajectory
- Transfer time to Venus of about 135 days
- Use of VGA and on-board propellant to achieve final orbits
- Heliocentric, elliptical orbits of 0.72×0.50 , 0.72×0.60 , 0.72×0.85 , and 0.72×0.95 AU in the plane of the ecliptic with orbit periods ranging from 5.9 to 7.5 months
- Earth/spacecraft/Sun aligned within one degree for about 2 weeks maximum
- No shadow periods for above orbits





IHS Launch Opportunity Options



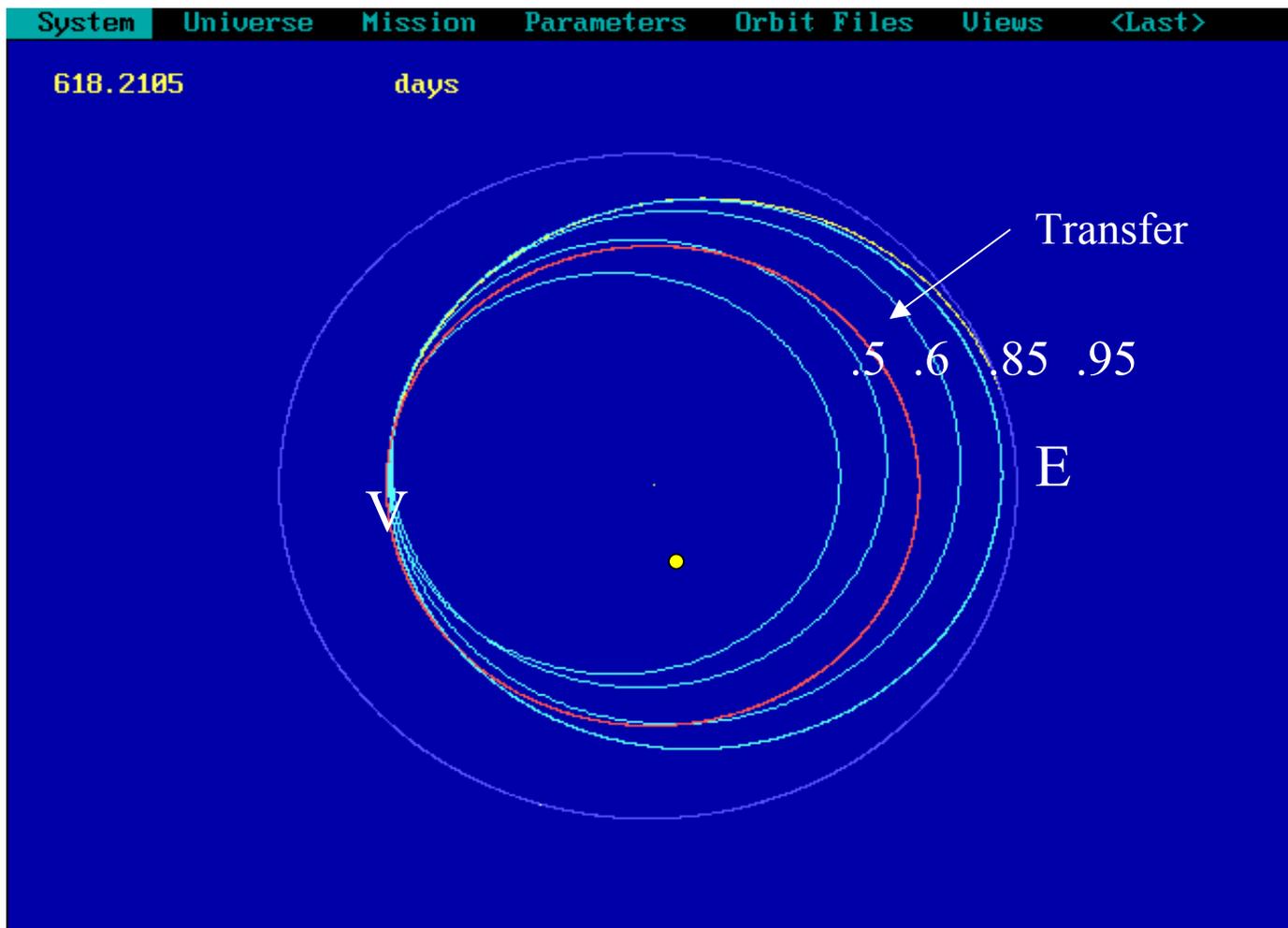
1st Opportunity:
12/25/08

2nd Opportunity:
08/13/10





IHS Orbit Pictorial





IHS Spacecraft Features

IHS spacecraft features include the following:

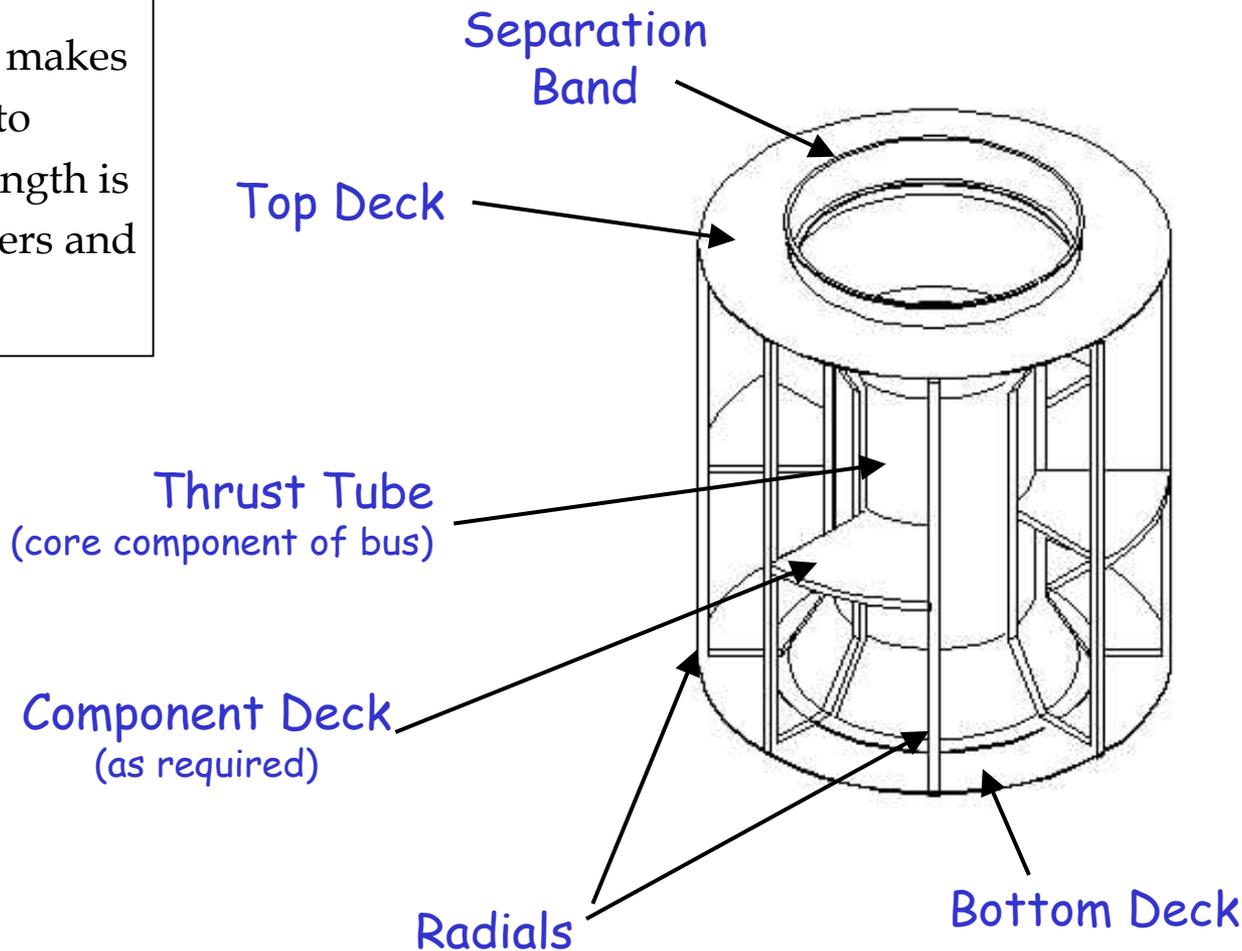
- Four nearly identical spacecraft spin-stabilized at 20 RPM with external solar array and radiator surface areas sized in accordance with orbit proximity to the Sun
- A mission unique spacecraft design that has some commonality with the RBM and IM flight elements
- Deployable instrument booms and wire antennas
- Body mounted solar arrays
- A Ka-band high gain antenna system on a despun platform for downlink of science data





IHS Mechanical Subsystem

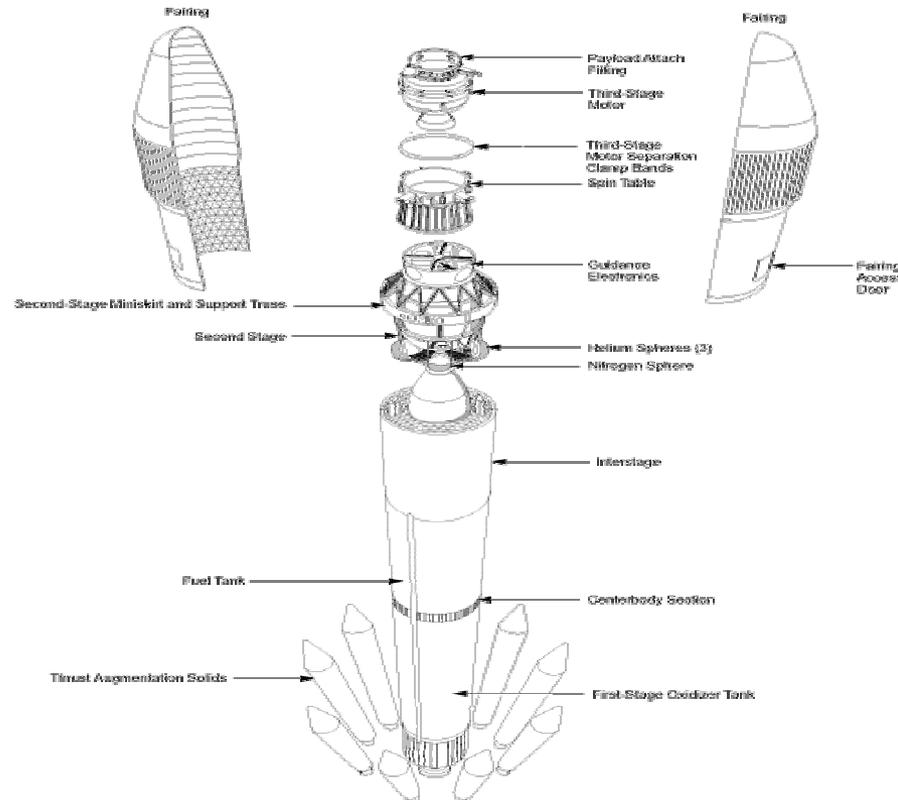
The IHS mechanical design makes use of composite materials to minimize launch mass. Strength is provided by selection of fibers and ply orientation.





IHS Launch Vehicle Evaluation

Launch vehicle options include the Pegasus XL and some Delta II configurations. The former requires four launches to accomplish the mission. The latter (Delta II 7925H) with a stretched fairing, as shown below, allows a single launch but mass margin must first be increased to an acceptable level.





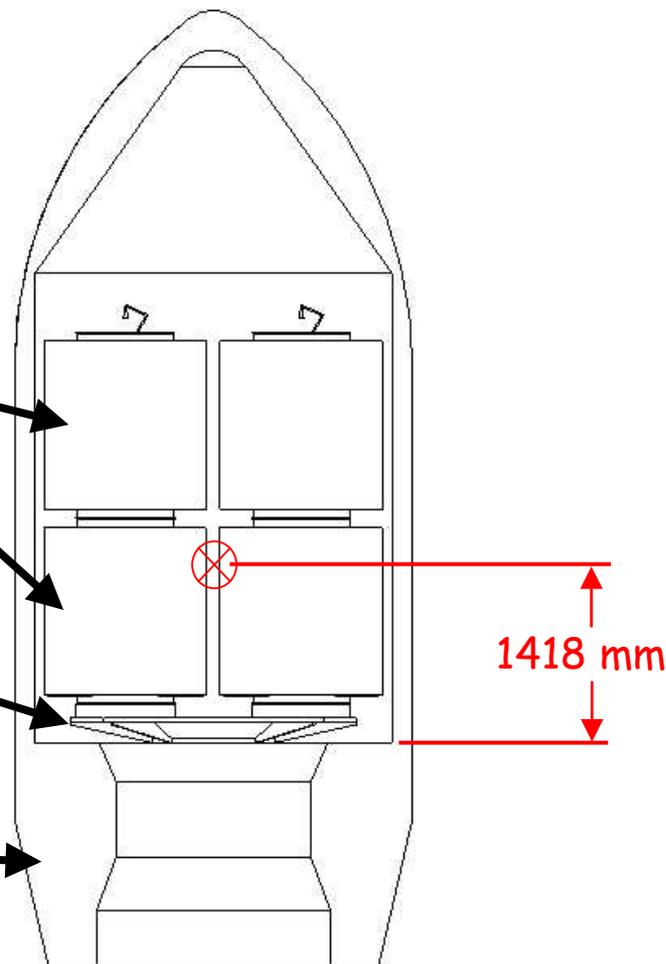
IHS Launch Configuration

*Note: Each satellite is approximately
1.2 m in diameter and 1.4 m high.*

Inner Heliospheric Sentinels
(2 stacks of 2 - 4 total)

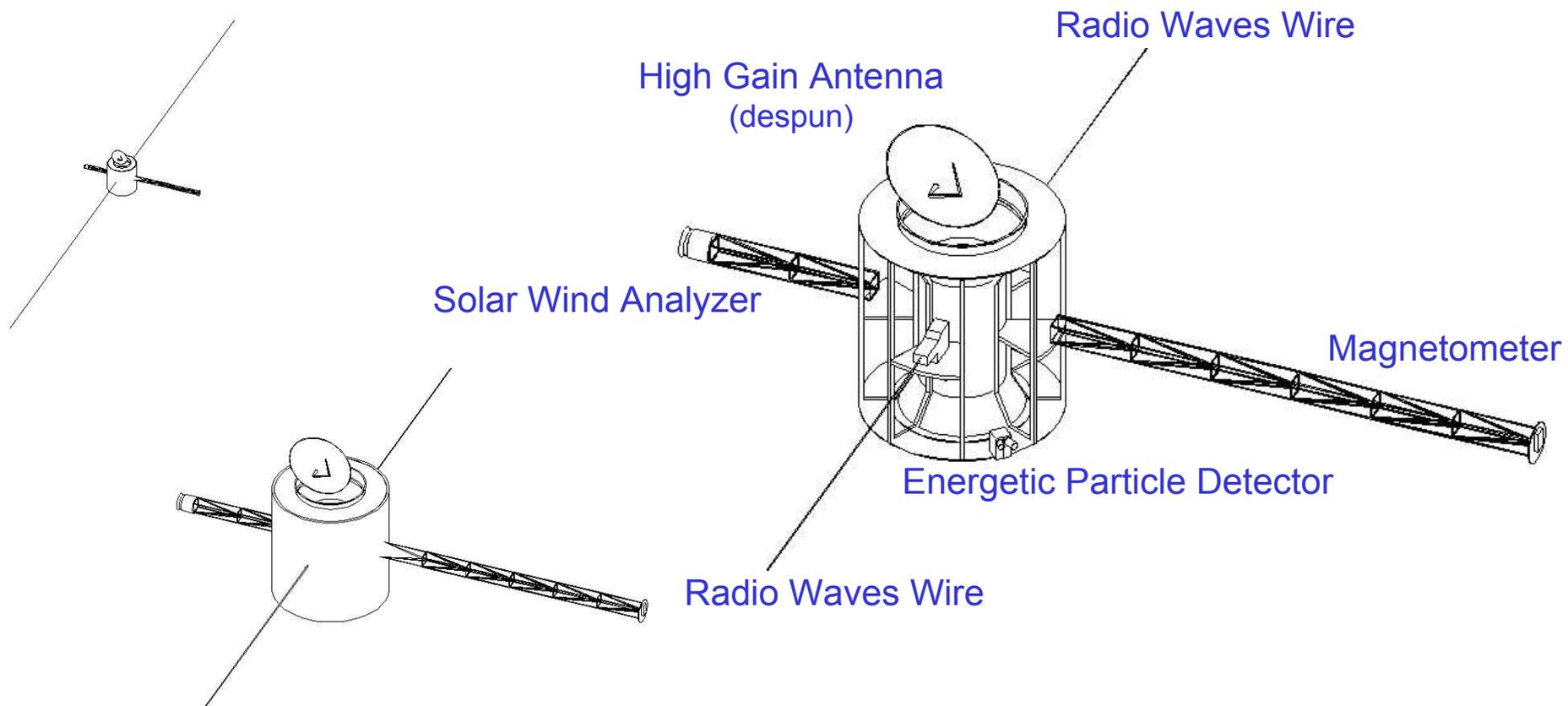
Double Payload Adapter Fitting

Delta II 7925H--3 m Stretched Fairing
(Third Stage)





IHS Orbit Configuration





IHS Power Subsystem

The IHS Power Subsystem is a SMEX type 28-volt direct energy transfer system that can support a peak load of 200 watts during RF transmission periods at the end of the nominal mission life with a cell temperature below 120° C. It consists of the following elements:

- Triple junction GaAs solar cells with coverglass covering up to 62% of the available spacecraft circumferential area
- A 33.5 ampere hour prismatic Li-ion battery module to handle launch and transfer orbit conditions
- Power supply electronics

Solar array degradation over the life of the mission due to UV exposure, ionizing radiation, thermal cycling, and system losses has been taken into account in the array sizing.





IHS Thermal Subsystem

A passive thermal design approach has been adopted that isolates the body mounted solar arrays from the spacecraft structure and mounts heat dissipating components to one of two space-viewing cylinder end caps.

Circumferential solar arrays and adjacent radiator surface areas are sized to provide required power and cell cooling in accordance with orbit proximity to the Sun. By judicious spacing of solar cells and optical solar reflectors, solar array temperatures are kept well below their 120° C upper limit.

Propulsion subsystem components are kept above 12° C to eliminate any concern over a potential freeze/thaw cycle. All other internal components are maintained between 0 and 40° C.





IHS Attitude Control Subsystem

The Attitude Control Subsystem (ACS) concept selected for the IHS satellites can accommodate the desired spin rate, pointing accuracy, and knowledge requirements as specified below:

- Maintain a stable spin rate of 20 RPM
- Keep the spin axis of each satellite normal to the ecliptic plane
- Provide attitude knowledge of 1 degree (1 sigma)





IHS Attitude Control Subsystem (continued)

The following complement of hardware is used in conjunction with propulsion subsystem thrusters to provide the functions required for attitude control of the satellites during deployment, spin-up, spin axis orientation, and any safe-hold condition:

- Coarse sun sensors
- Fan-shaped digital sun sensors
- Simple slit star camera
- Nutation dampers





IHS Propulsion Subsystem

The Propulsion Subsystem concept chosen for each satellite is a mono-propellant hydrazine system sized to correct for launch vehicle dispersion and provide the thrust for the planned Venus swing-by.

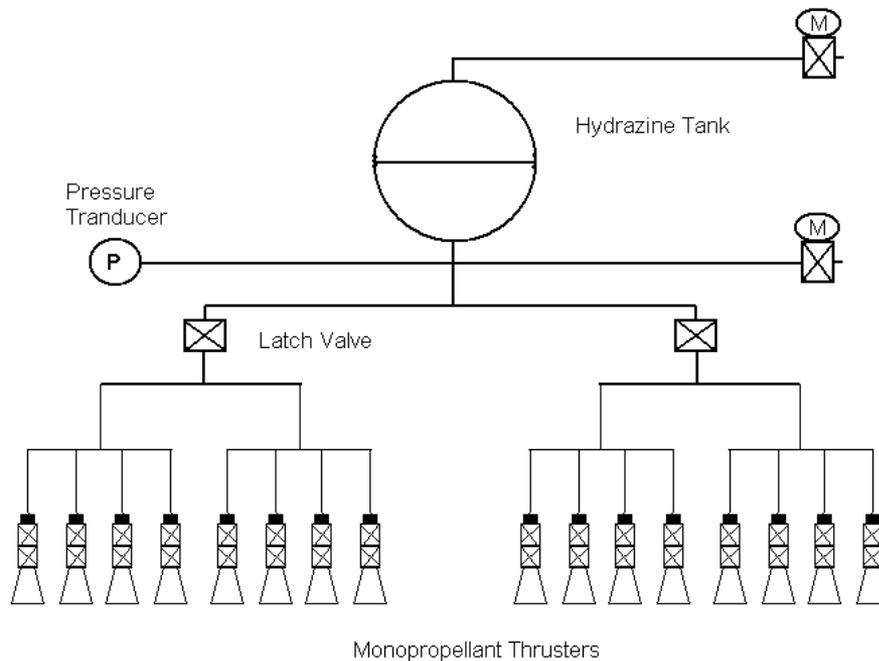
The total DV requirement was estimated to be about 100 m/s per satellite.

A schematic representation of the proposed Propulsion Subsystem is shown in the chart that follows.





IHS Propulsion Subsystem Schematic





IHS Command and Data Handling Subsystem

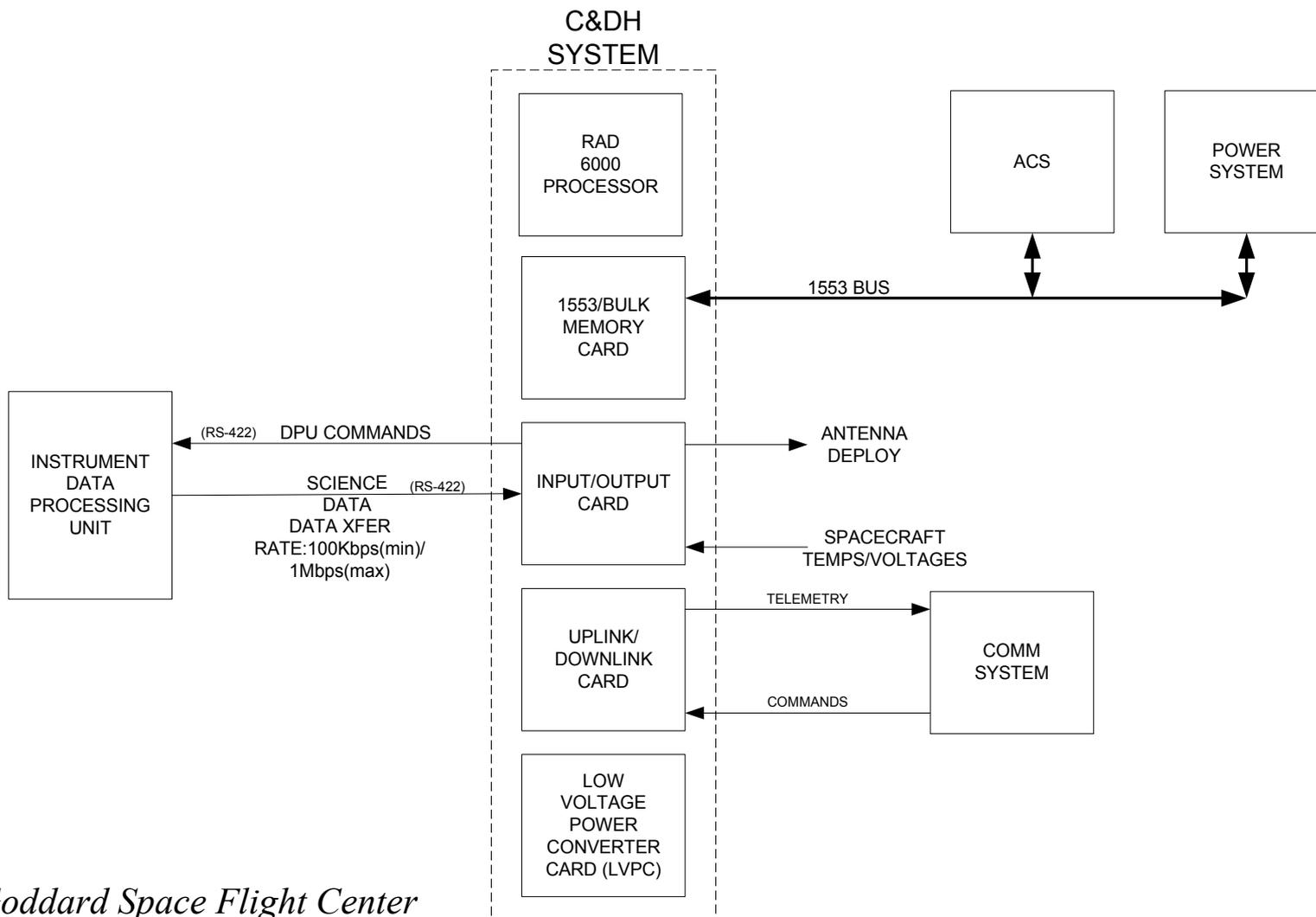
The C&DH Subsystem provides the following functions and services:

- Science data ingest from the Instrument DPU
- Storage of science and spacecraft housekeeping data
- Processing of commands and telemetry
- Interfaces with power and attitude control subsystems
- Acquisition of analog data for temperature and voltage monitoring





IHS C&DH Subsystem Schematic





IHS Communications Subsystem

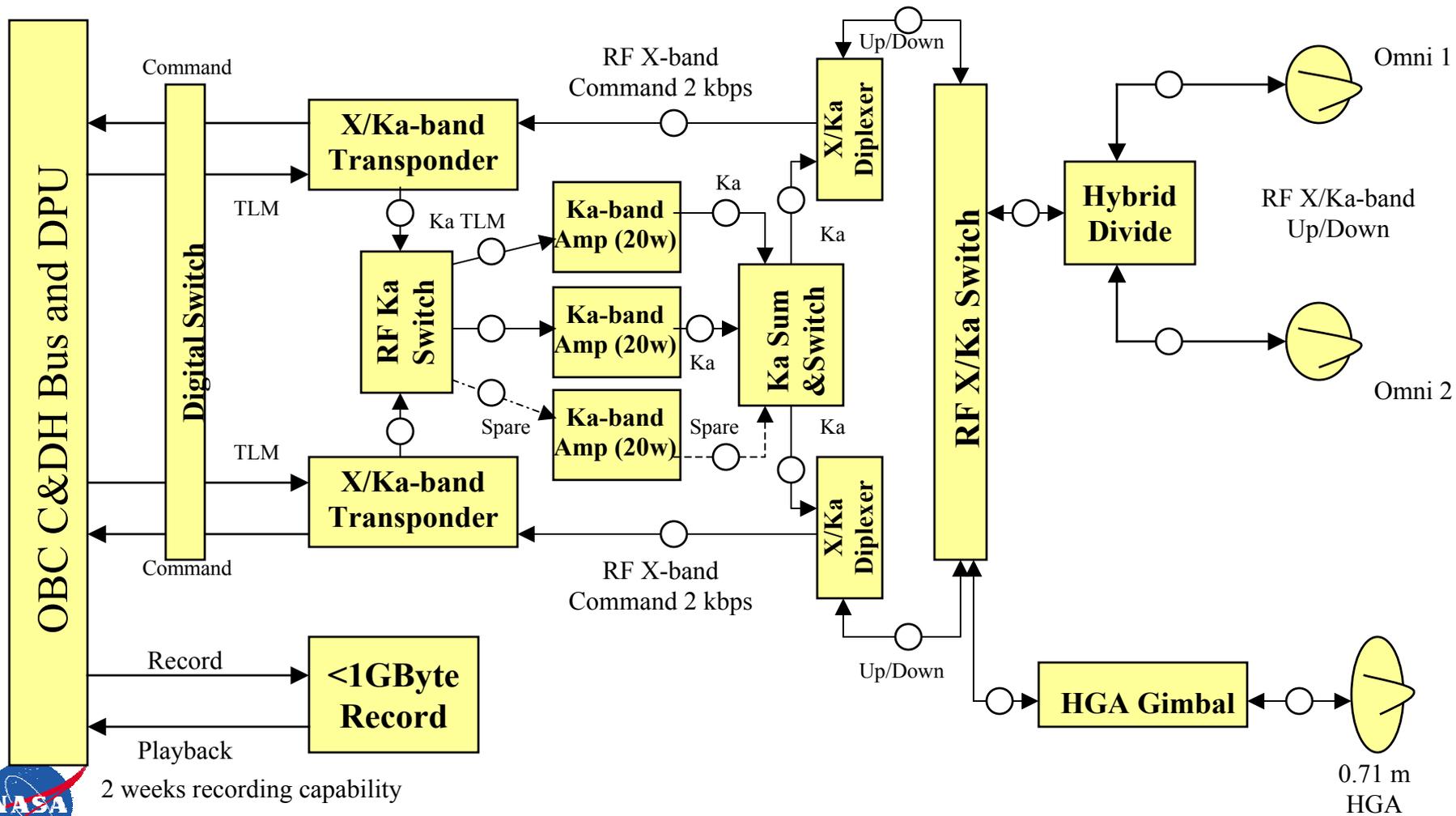
The Communications Subsystem for each satellite, shown in the schematic that follows, has the following features:

- A Ka-band high gain antenna on a despun platform for science data downlink
- X-band omni-directional antennas for commanding
- Redundant X-band and Ka-band transponders
- Provisions for contingency modes





IHS Communications Schematic





IHS Ground System

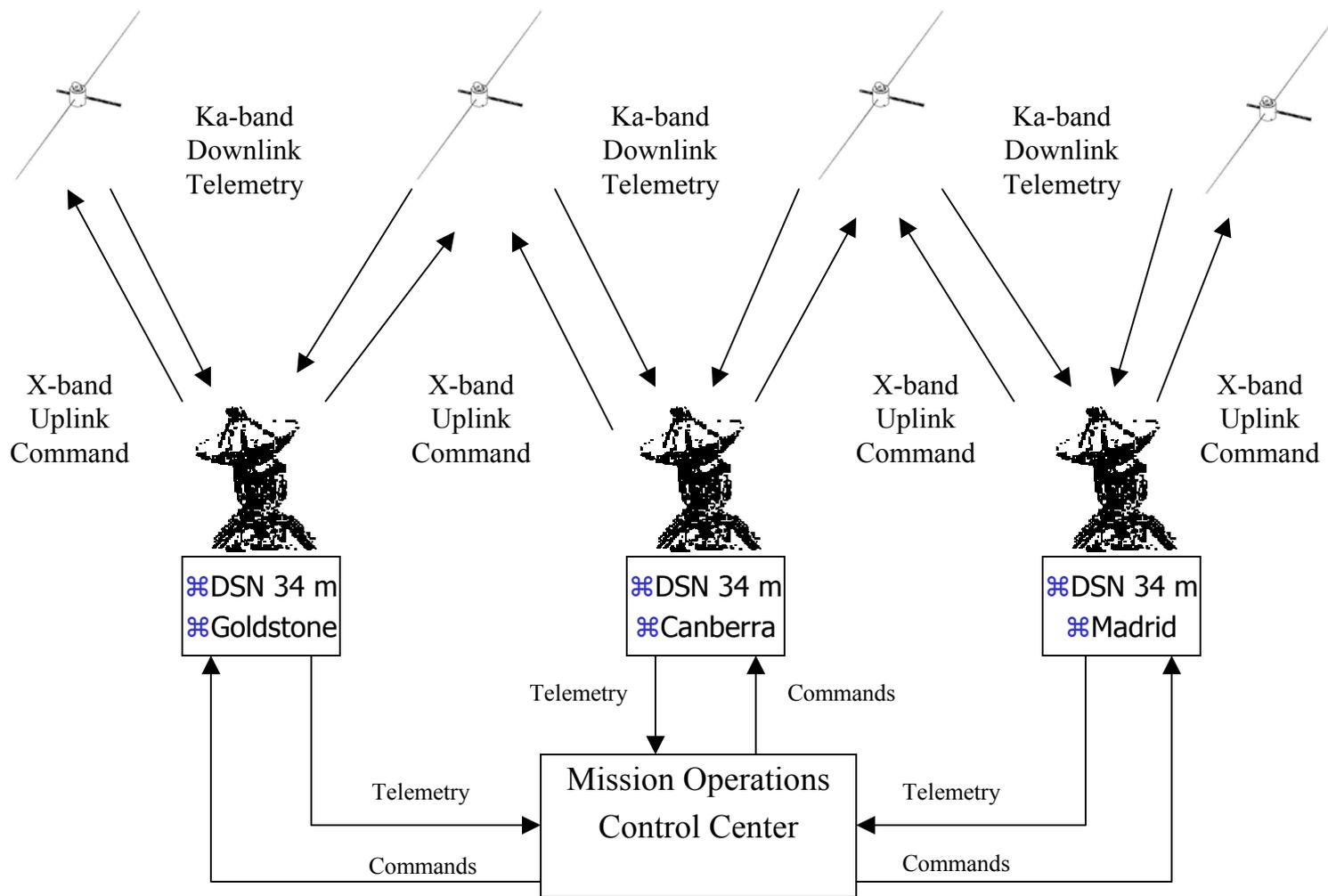
The proposed Ground System accommodations take advantage of existing infrastructure and include the following features:

- Three equally spaced 34 m DSN ground stations at Goldstone, Canberra, and Madrid
- Science data compressed, stored, and dumped at 100 kbps for each satellite
- Command rate of 2 kbps
- Ranging twice per week
- Data stored at each ground site and delivered via ftp to the Mission Operations Control Center





IHS Ground System Concept





IHS Mission Operations

A Mission Operations concept has been adopted that encourages automation and makes use of commercial-off-the-shelf (COTS) products. Salient features include the following:

- A multimission operations center staffed by a small Flight Operations Team to support the fleet of 4 satellites
- Automated mission operations, wherever possible, using a COTS command and control system
- One contact per day per spacecraft
- Hot backups for critical command and telemetry processors
- On-board recording capacity for 2 weeks of science/housekeeping data during solar occultation periods that occur three times a year
- Lights out operations after an orbital routine is established





IHS Science Data Accommodations

IHS science data accommodations include the following provisions:

- Recovery of 95% of all data
- Generation of Level 2 science data products and transfer to an existing system like NSSDC within 48 hours
- Storage of raw telemetry data at the DSN sites for 7 days
- Real time routing of critical housekeeping data





IHS Mission Specific Technology

The IHS mission concept incorporates new technologies that are expected to be available well before the target implementation phase. These items include the following:

- High-efficiency, triple-junction, GaAs solar cells for power generation
- Li-ion battery modules for energy storage
- Simple-slit star camera for spin axis orientation
- X-band omni antennas for command uplink
- Ka-band high gain antenna system for telemetry downlink





IHS Preliminary Risk Assessment

During IHS concept definition, a number of risk areas were identified and are listed below. Further study will be required to fully assess these risks, their potential impact, and mitigation strategies.

- The present launch mass margin is inadequate for four satellites on a medium class ELV.
- Clearance for the high gain antenna systems in a stacked launch configuration and deployment issues require detailed evaluation.
- Fairing access to allow on-stand off-loading of propellant from the stack of satellites in the event of an emergency will require special provisions.
- Although conservative assumptions have been made, availability of mission specific technologies must be reviewed at regular intervals.





IHS Study Recommendations

- Evaluate options for reducing the mass of each satellite.
- Develop a detailed operations timeline that includes separation, deployment, and cruise to the final mission orbit and determine system performance requirements including power, attitude control, propulsion, and communications for each satellite.
- Study methods for increasing the gain of the Ka-band system to reduce EIRP and to allow operation at lower elevation angles.
- Identify process developments that simplify building and testing of multiple research grade instruments and spacecraft.





Acronyms

ACE	Advanced Composition Explorer
ACS	Attitude Control Subsystem
AU	Astronomical Unit
BOL	Beginning Of Life
C&DH	Command and Data Handling
COTS	Commercial Off-The-Shelf
DPU	Digital Processing Unit
DSN	Deep Space Network
DSS	Digital Sun Sensor
EOL	End Of Life
FSS	Far Side Sentinel
GaAs	Gallium Arsenide
GSFC	Goddard Space Flight Center





Acronyms

IHS	Inner Heliospheric Sentinels
IMDC	Integrated Mission Design Center
JPL	Jet Propulsion Laboratory
LWS	Living With a Star
MOCC	Mission/Multimission Operations Control Center
RPM	Revolutions Per Minute
STEREO	Solar Terrestrial Relations Observatory





Far Side Sentinel



J E T P R O P U L S I O N L A B O R A T O R Y



FSS Mission Profile

Description: Continuous observation of activities on the far side of the Sun in order to track the full evolution of space weather disturbances and enable improved space weather forecasting

Instruments: A combination of remote sensing and in-situ instruments including a doppler magnetograph, an EUV imager, a magnetometer, solar wind plasma and energetic particle detectors, and an optional radio science experiment

Spacecraft: A three-axis stabilized, solar-pointed spacecraft with a propulsion system for orbit maneuvers

Launch Date: January or November 2009

Mission Life: 2 years on station with an optional 3-year extension of mission operations as resources allow

Orbit: Solar far side drift orbit

Space Access: Launch from ETR on a Medium-Lite Class ELV

Key Technologies: Enhancing technologies at the subsystem or component level available by 2006 via the X2000 program or equivalent





FSS Mission Timeline

The following serial time spans are assumed for mission planning:

- Extended pre-formulation and formulation phase instrument, spacecraft, and ground system accommodation studies to match available funding
- 12 months for conclusion of project formulation and definitization prior to approval
- 2.5 to 3.3 years from approval to launch readiness dependent upon LWS program funding profile
- January or November 2009 launch
- 12 month cruise to operational drift orbit
- 2 years for baseline mission operations
- 3-year mission extension (option for evaluation)





FSS Mission Objectives

The FSS mission employs a three-axis stabilized spacecraft and its complement of remote sensing and in-situ instruments to make continuous solar observations of the far side of the Sun. Specific mission objectives include the following:

- Search for disturbances forming in the Sun's convection zone and rising to the photosphere
- Track the evolution of these disturbances to enable daily/weekly space weather predictions
- Characterize coronal mass ejections (CMEs)
- Measure coronal magnetic fields
- Obtain global magnetic boundary conditions to model the three-dimensional structure of the heliosphere





FSS Instrument Complement

The FSS instrument complement consists of the following types of measurement devices:

- (1) Doppler magnetograph for remote sensing of photospheric magnetic and velocity fields
- (2) EUV imager for remote sensing of the solar corona
- (3) Magnetometer for in-situ magnetic field measurements
- (4) Faraday cup for in-situ solar wind plasma measurements
- (5) Energetic charged particle detector for in-situ measurements
- (6) Optional radio science experiment using the telecommunications system

All the instruments above use proven technology and have strong flight heritage.





FSS Baseline Instrument Complement

Type/Classification	Mass	Power	Data
	(kg)	Avg/Peak (W)	Storage (Mb/day)
Doppler Magnetograph	14	30	290
EUV Imager	14	27.5	150
Magnetometer	1.5	2	1.6
Faraday Cup	1.5	2	0.2
Energetic Particle Detector	0.3	0.2	1.6
Total	31.3	61.7	443.4
Radio Waves Experiment Option*	17.8	115	
* Part Of Spacecraft Communications Subsystem			



FSS System Synopsis

- **Launch Date:** 2009 from Eastern Test Range
- **Technology Cutoff:** 2006
 - * JPL X2000 technology development program used as a reference
 - * Schedule margin added to X2000 program first delivery for risk reduction
- **Payload Accommodations**
 - * Payload mass (31.3 kg), power (61.7W), and data volume (472 Mb/day)
 - * Radio science desired
 - * Twice weekly return of science data (nominal)
 - * Data storage for 10 to 20 day communications blackout period ($<2^{\circ}$ SEP angle)
- **3-Year Mission Life Time:**
 - * 1-year transit, 2-years on station
 - * 3-year mission extension
- **Custom Bus Design**
 - * Consumables sized for 6 years
 - * RSDO BCP-600 examined as industry alternative

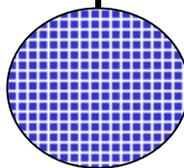




FSS System Block Diagram

ELECTRICAL POWER

SOLAR ARRAY



Li-ion BATTERY

X2000 POWER ELECTRONICS

COMMAND & DATA HANDLING

THERMAL SUBSYSTEM

CPU

LV POWER SUPPLY

BACKPLANE

INTERFACE

LV POWER SUPPLY

PROPULSION ELECTRONICS

MEMORY

COMMUNICATION INTERFACE

HYDRAZINE

FUEL FILTER

REG

INSTRUMENT

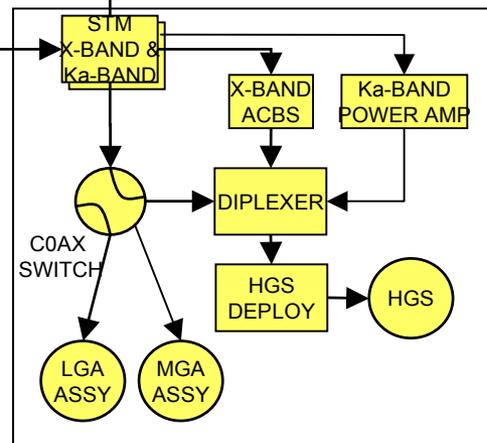
5 INSTRUMENTS

NOTE: RADIO INSTRUMENT INCLUDED IN TELECOMM

Sun Sensor

STAR TRACKER

IRU



0.2 lb (X8)

1.0 lb (X8)

PROPULSION

PAYLOAD

ATTITUDE CONTROL

TELECOMMUNICATIONS





FSS Mass Summary

FLIGHT SYSTEM ELEMENT	MASS (kg)
INSTRUMENTS	31.3
ATTITUDE CONTROL	4.6
COMMAND AND DATA	5.9
POWER	12.3
PROPULSION	10.1
STRUCTURE	37.5
S/C Adapter	6.0
CABLING	13.0
TELECOMM	17.4
THERMAL	7.9
SUBTOTAL	146.0
MASS CONTINGENCY (30%)	43.8
SUBTOTAL with CONTINGENCY	189.8
PROPELLANT & PRESSURANT	20.0
INJECTED MASS	209.8

DELTA II 7326 (Baseline)

LAUNCH VEHICLE CAPABILITY*	435.0
LAUNCH VEHICLE MARGIN	107% (225.2 kg)

* Launch Vehicle Adapter Included

ATHENA II/Star 48 (Option)

LAUNCH VEHICLE CAPABILITY	200.0
LAUNCH VEHICLE MARGIN[†]	-0.12% (-27.8 kg)

[†] 18 kg Allocation For Launch Vehicle Adapter Added To Injected Mass

Stated margins are based on an injected mass that already includes contingency.





FSS System Power Summary*

• Assumptions

- Solar Powered with Battery Backup
- X-2000 Power Electronics
- Maximum Solar Distance of 1.2 AU
- Ultra-Flex Solar Array
- Triple-Junction Solar Cells
- Designed for High Power Mode
- 30% Contingency

• Power Modes

- Mode 1: Dual telecom for radio science and communications
- Mode 2: Full science and no telecom
- Mode 3: Power usage during trajectory control maneuvers (TCM)
- Mode 4 :Power in normal cruise before science data taking
- Mode 5: Launch mode used to size the battery.

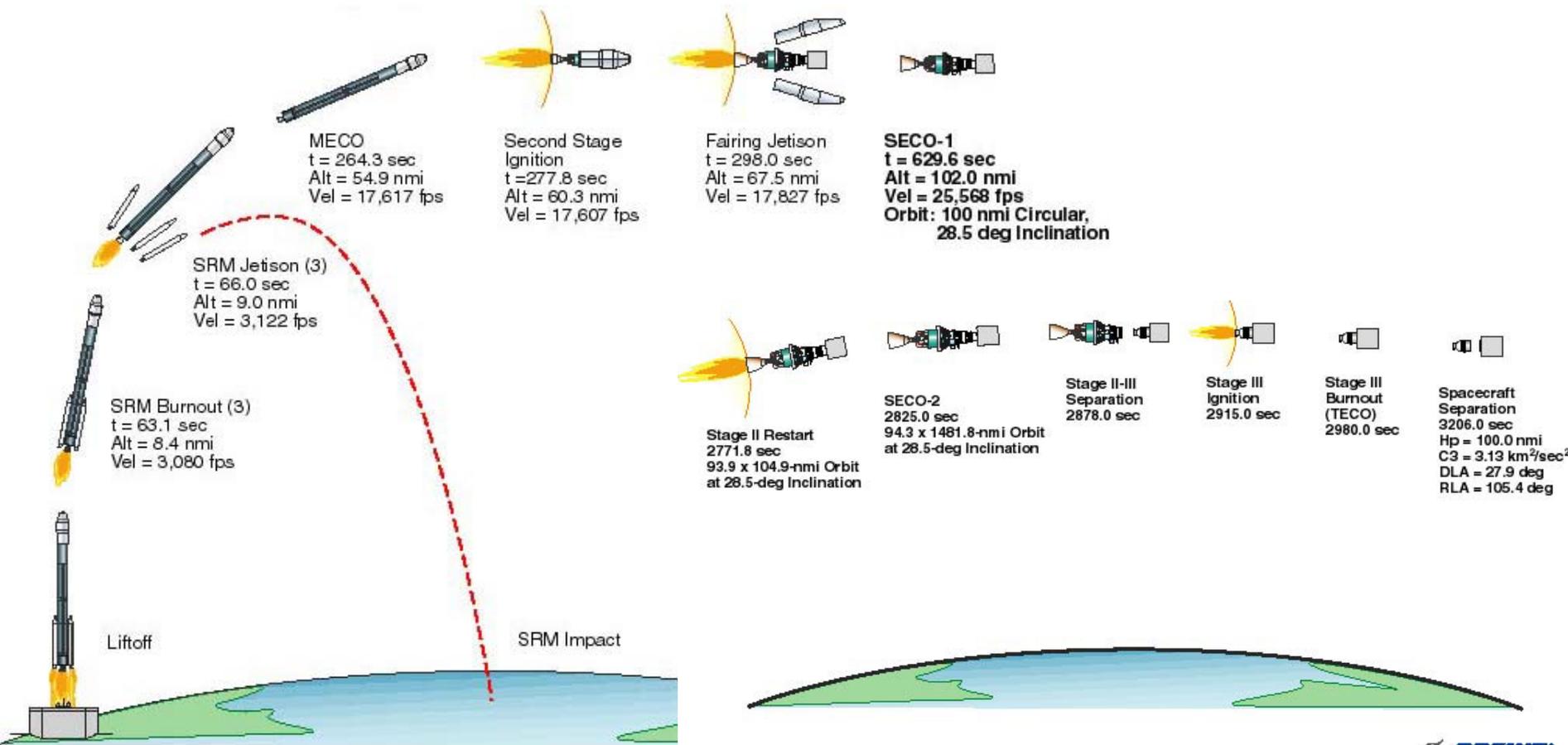
FLIGHT SYSTEM ELEMENT	Power* (W)
INSTRUMENTS	4.2
ATTITUDE CONTROL	4.5
COMMAND AND DATA	11.5
POWER	19.0
PROPULSION	37.3
TELECOMM	56.8
THERMAL	32.2
SUBTOTAL	165.5
POWER CONTINGENCY (30%)	49.7
SUBTOTAL with CONTINGENCY	215.2

* Mode 3 is the high power mode for the system even though instrument power is minimal.





Delta II 7326 Launch Profile*



* From DS-1 Mission Book, 24 Oct 1998, Boeing Space Systems



FSS Orbit Trajectories

- Venus Flybys Used to Minimize ΔV and C_3 Requirements**

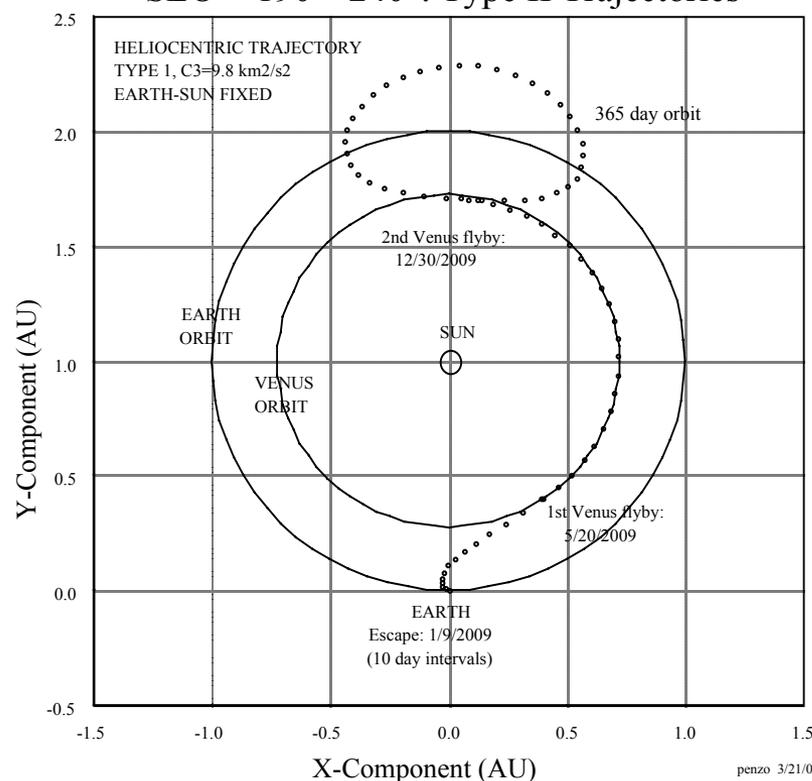
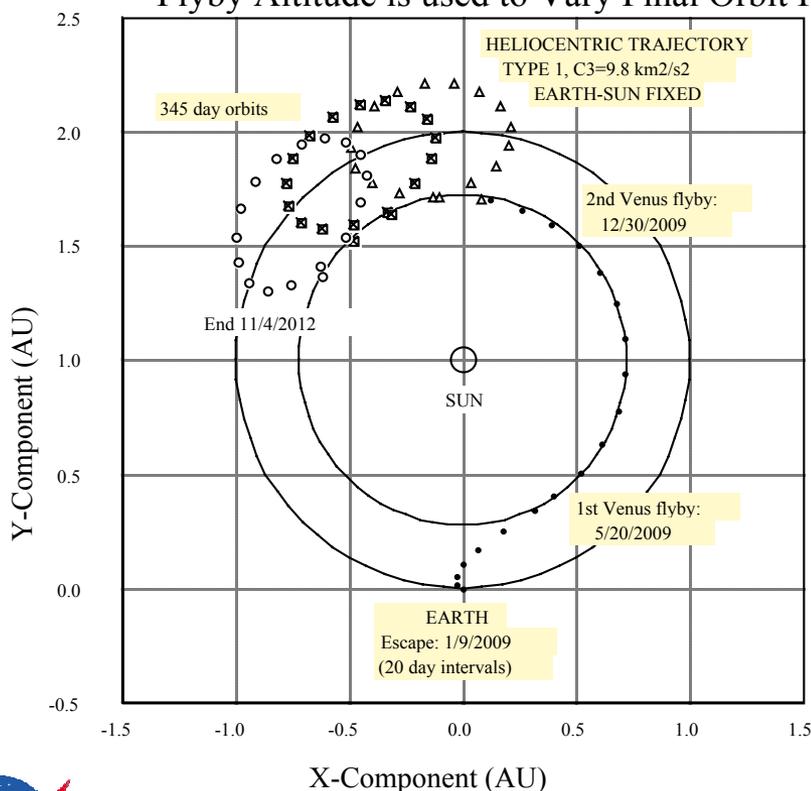
- * Purely Ballistic Flight Path
 - No deterministic maneuvers
- * Time to “First” Venus Flyby ~ 4 mos.
- * Flyby Altitude is used to Vary Final Orbit Period

- Orbit Options Illustrated Below**

- * Stationary Orbit (365-day Period)
- * Drift Orbit Preferred (< 365 -day Period)

- Initial Orbit Phasing**

- * $SEO^\ddagger = 120^\circ - 170^\circ$: Type I Trajectories
- * $SEO = 190^\circ - 240^\circ$: Type II Trajectories



penzo 3/21/00

$\ddagger SEO \equiv$ Sun-Earth-Orbit Centroid Angle





FSS Orbit Pictorial

- **Launch Date: 2009**

- * 20-day launch window
- * $C_3 = \sim 14 \text{ km}^2/\text{s}^2$ (maximum for window)
- * Delta II 7326 (or Athena II /S48)

- **Platform Location**

- * Station spacecraft behind Sun within 30° of the extended Earth-Sun line

- **Spacecraft - Sun Relative Velocity**

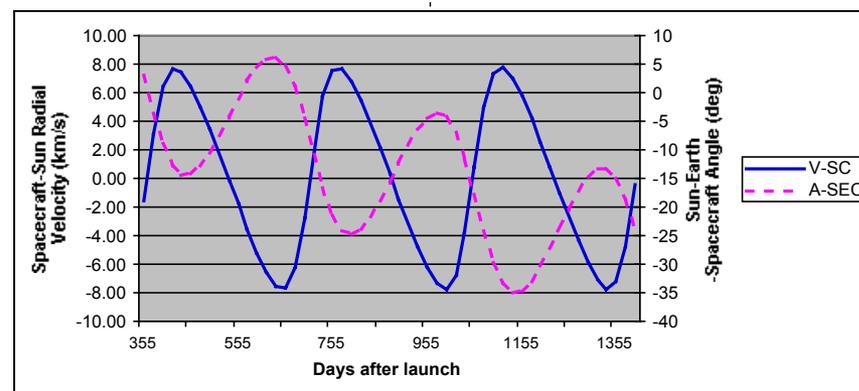
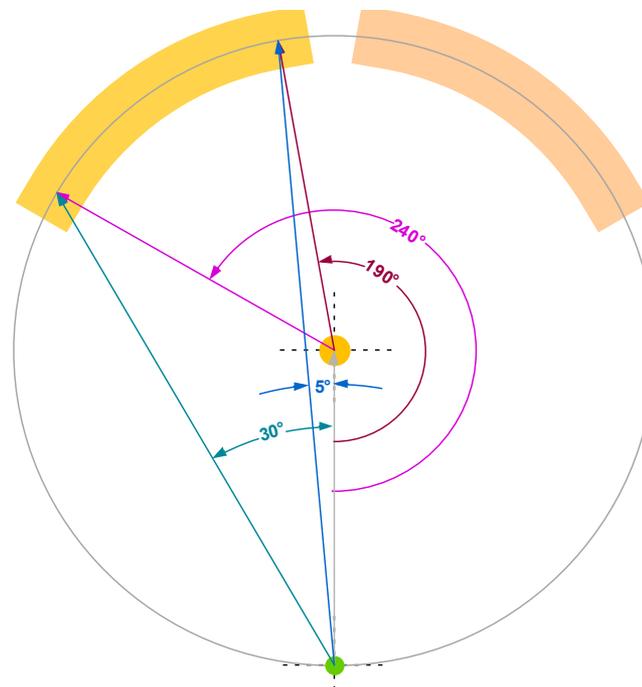
- * Minimize time $V_{\text{rel}} > 4 \text{ km/s}$

- **Time On Station**

- * 2 to 5 years

- **DSN Data Return Conjunction Constraint**

- * $\text{SEP}^\dagger = 2^\circ$ to 5°
 - Depends on 1- or 2-way coherent link capability
- * 2° used for reference mission





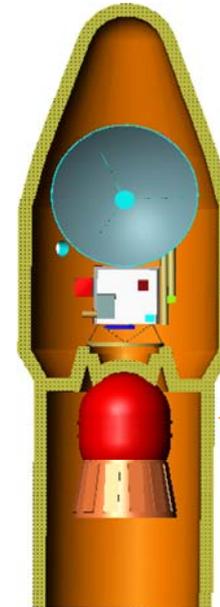
FSS Launch Configuration

Delta II Assumptions

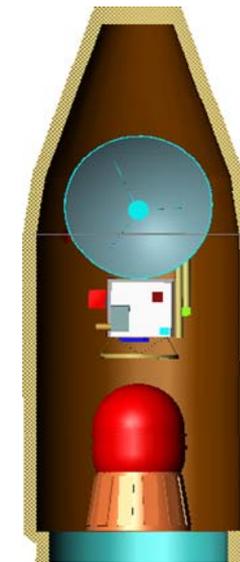
- Delta II 7326 (Baseline L/V)
- 435 kg Capability at $C_3=14 \text{ km}^2/\text{s}^2$
 - * 5% performance derating taken
 - * L/V adapter accounted for in performance
- 9.5 ft. Fairing Size
- Spinning Injection
- Accommodates up to 2.54 m Antenna
- Special Considerations: None

Athena II Assumptions

- Athena II+Star 48 Kickstage
- 200 kg Capability at $C_3=14 \text{ km}^2/\text{s}^2$
 - * 15% performance derating taken
 - * L/V adapter booked against spacecraft
- 10 ft. Fairing Size
- Spinning Injection
- Accommodates up to 1.98 m Antenna
- Special Considerations:
 - * Fairing **NOT** flight qualified



Final Stage of Delta II 7326



Additional Kickstage Added to Athena II

Payload Fairings **NOT** to Relative Scale





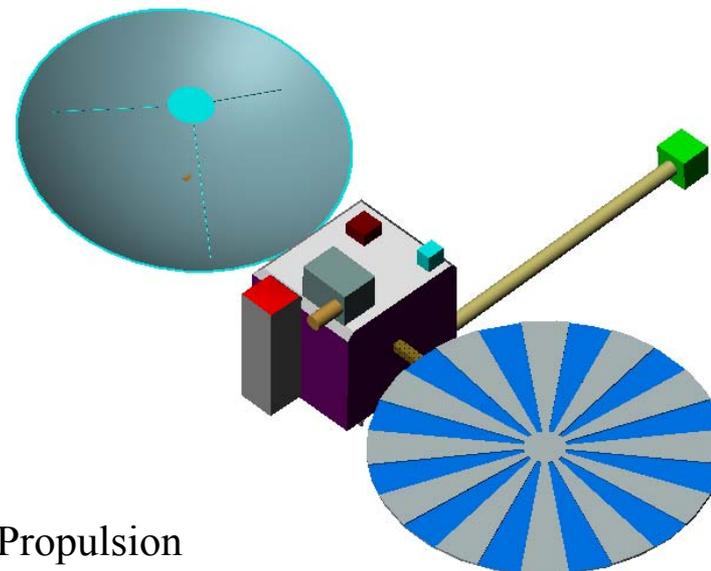
FSS Orbit Configuration

• Assumptions

- 30% Mass Contingency
- 30% Power Contingency
- 3-Year Primary Mission Lifetime
- Allow for Solar Conjunctions
- Accommodate Extended Mission

• Key Characteristics

- Payload
 - * 5 instruments
 - * 450 Mbits/day (average)
- Attitude Control
 - * 3-axis stabilized
- Power
 - * X2000 power electronics
 - * Ultra-Flex solar array (1.3 m²)



- Propulsion
 - * Hydrazine for 6 years
- Command and Data Handling
 - * 10 Gbit data storage
- Telecommunications
 - * Dual band (X/Ka)
 - * 2.0 m antenna (baseline)
 - * 64 kbps nominal downlink rate





FSS Mechanical Subsystem

Assumptions

- Composite structure with a TRL of 6
- 2 m diameter high gain antenna
- 2 m deployed magnetometer boom

Design

- Custom bus design with a mass fraction of 26% (cabling is at 9%)
 - * Deployable non-articulated antenna
 - * Deployable non-articulated solar array
 - * Telecom mounted on a side pallet with radiators to dissipate heat
- RSDO BCP-600 Bus Evaluated
 - * Deployable and articulated antenna
 - * Needs a heavier structure due to the solar arrays and the telecom package

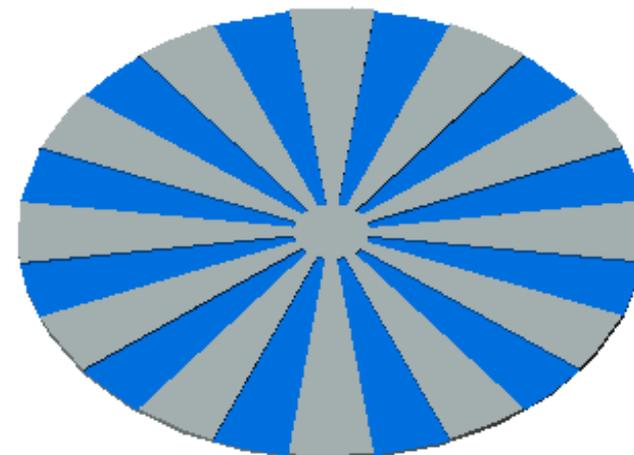




FSS Mechanical Subsystem

Assumptions

- Power sized for maximum sun distance of 1.2 AU
- Solar array to be qualified to 180 °C for Venus
- Battery sized for 3 hour launch window at 80% Depth of Discharge



Design

- X2000 First Delivery for Power Electronics
- Deployable Ultra-Flex Solar Array
 - * Triple junction GaAs/InP₂/Ge solar cells @ 24% efficiency
 - * Array area of 1.3 m² required
 - * Array sized for 15 degree cosine loss
- Battery
 - * 15 Ampere-hour Li-ion battery with a total of 432 W-hr capacity





FSS Thermal Subsystem

Assumptions

- Distance from the sun is 0.7 to 1.2 AU
- Heaters necessary for instruments from start of mission to science data taking

Design

- Passive design with louvers necessary for telecom power dissipation
- Multilayer insulation
- Variable emissivity surface (permitted by 2006 technology cutoff)
- Heaters and radiators for the science instruments





FSS Attitude Control Subsystem

Assumptions

- TRL of 5 with a technology cutoff of 2006
- Pointing accuracy of 90 arcsec or 0.025 degrees

Design

- 3-axis stabilized
- Equipment
 - * Sun Sensors
 - * Star Trackers (X2000 program)
 - * Inertial Reference Units
- MCM interface electronics
- Pointing
 - * Accuracy = 70 microradians = 0.0035 degree
 - * Knowledge = 60 microradians = 0.003 degree
 - * Stability = 20 microradians/sec = 0.001 degree/sec
 - * Slew Capability = 1 degree/sec





FSS Propulsion Subsystem

Assumptions

- Delta V requirement of 150 m/s; ACS propellant estimate at 6 kg
- Selected redundancy in thrusters
- Integrated spacecraft dry mass of 190 kg

Design

- Hydrazine blow-down system
- Eight 4.45 N (1 lb) thrusters and eight 0.9 N (0.2 lb) Minimum Impulse Thrusters (MIT)
- 20 kg of propellant based on spacecraft mass and ACS propellant
- Key components used
 - * Lightweight pressure transducer for mass savings, radiation tolerance, and increased precision in the measurement
 - * Lightweight filter for mass savings
 - * Composite Pressure Vessel (COPV) propellant tank for mass savings
 - * Lightweight service valve for mass savings
 - * Minimum Impulse Thrusters (MIT) for finer control (a minimum impulse bit of 3 mN-sec) and elimination of reaction control wheels





FSS C&DH Subsystem

Assumptions

- Storage of science data during blackout periods
- 64 kbps downlink data rate with 1 Mbps possible from the instruments

Design

- Block redundant
- X2000 program first electronic delivery
- 10 Gbit data storage requirement
 - * 450 Mbps(+) per day with possible 20 days of occlusion
 - * 1 Gbit buffer storage for downlink catch up





FSS Communications Subsystem

Assumptions

- Dual frequency system
- Maximum distance of 2.3 AU to Earth
- Data rate of 64 kbps with a 34 m beam wave guide DSN antenna
- Radio science
- 2° data transmission conjunction constraint for the Sun

Design

- X-band/Ka-band
 - * During cruise mode only X-band will be used
- Radio science experiment needs both X-band and Ka-band
 - * Radio science is major design driver
 - Additional 80 watts of RF power required for X-band link to 34 m DSN station





FSS Ground/Data System

• Spacecraft Data

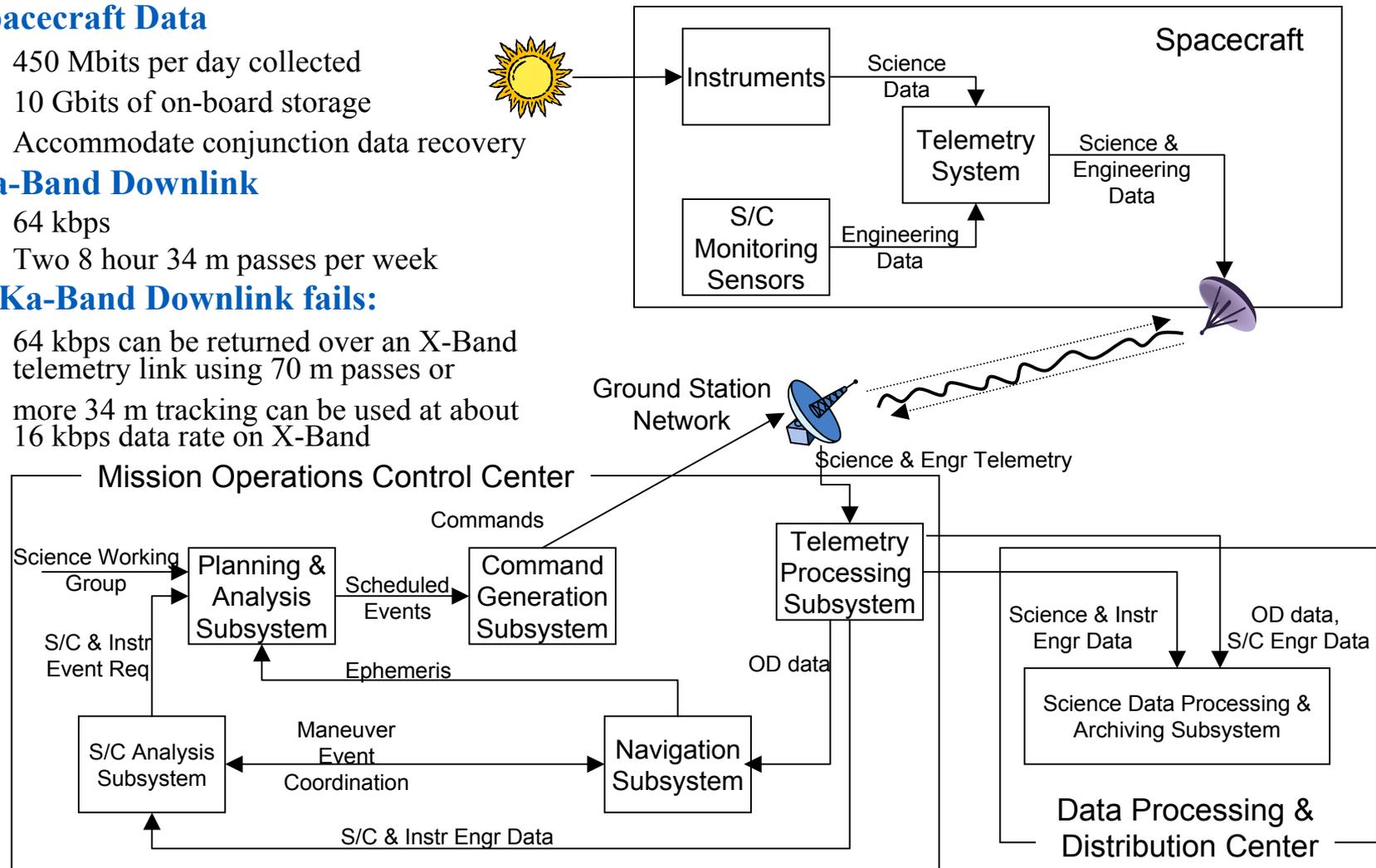
- * 450 Mbits per day collected
- * 10 Gbits of on-board storage
- * Accommodate conjunction data recovery

• Ka-Band Downlink

- * 64 kbps
- * Two 8 hour 34 m passes per week

• If Ka-Band Downlink fails:

- * 64 kbps can be returned over an X-Band telemetry link using 70 m passes or
- * more 34 m tracking can be used at about 16 kbps data rate on X-Band





FSS Mission Operations

• Assumptions

- DSN 34 m beam wave guide as primary net
- Data return via 64 kbps Ka-band downlink (normal science operations)
- Nominal coverage with two 8-hour passes/week
- X-band link data return in event of Ka-band failure

• Data Return Strategy Around Solar Conjunction

SEP Above 5 Degrees	SEP Between 2 and 5 Degrees	SEP Below 2 Degrees	SEP Between 2 and 5 Degrees	SEP Above 5 Degrees
N Days 450 Mbits Collected Per Day Two 8 hour X- Band Telemetry Passes (64 Kbps) Per Week	10 to 20 Days 450 Mbits Collected Per Day One 4 hour Ka- Band Telemetry Pass (64 Kbps) Every Other Day and Two 8 hour passes Per day for Radio Science	10 to 20 Days 9 Gbits Collected No Telemetry Passes Two 8 hour passes Per day for Radio Science	10 to 20 Days 450 Mbits Collected Per Day One 4 hour Ka- Band Telemetry Pass (64 Kbps) Every Other Day To Play Back Current Data and Two 8 hour passes Per day for Radio Science	7 Days 450 Mbits Collected Per Day One 8 hour X- Band Telemetry Pass (64 Kbps) Per Day to Play Back Current Data and 9 Gbits Collected During Conjunction





FSS Technology Investment

- Technology cutoff: 2006
- Technology investment handled by X2000 program
 - * FSS schedule more conservative than X2000 delivery schedule
 - Technology program in place meets needs of FSS prior to 2006
- New technology items included in mission concept
 - * Ultra-Flex solar array
 - * Triple-junction GaAs solar cells
 - * Li-ion battery
 - * Power electronics
 - * Variable emissivity surfaces
 - * COPV propellant tank
 - * Light weight pressure transducer, filter, and service valve
 - * Minimum impulse thrusters
 - * Star tracker
 - * Ka-band antenna system





FSS Study Options

• Trajectory Options

- Ballistic and low-thrust options (solar electric propulsion) considered
 - * Ballistic option includes primary payload and Ariane-5 ASAP scenarios
 - Ballistic/primary payload mode selected as baseline
 - * Low-thrust option delivered platform to “stationary” position
 - Based on DS-1 ion drive

• Launch Vehicle Options

- Ariane-5 ASAP, Athena II+Star 48, Delta II 7326 examined for compatibility
 - * Delta II offers the only positive performance margin (+225 kg)

• Flight System Options

- Industry bus (Ball Aerospace BCP-600) examined as example of “today’s technology”
 - * Major modifications required to meet mission needs
- Mars Micromission bus examined for Ariane-5 option
 - * Final spacecraft design exceeded Ariane-5 ASAP mass and volume allocation

• Orbit Options

- Stationary and drift orbits examined
 - * Leading and lagging phasing considered





FSS Study Recommendations

Programmatic

- Launch phasing with other LWS program elements needs definition
- Duration of “formulation” and “implementation” phases needs clarification

Flight System

- Radio science is MAJOR design driver
 - * Requirements for margin on carrier need to be defined and traded against link margin requirement
 - * Occultations are required but only occur early in mission
- Evaluate design for mass reductions to fit Athena II class launch vehicle
 - * Possible \$15M reduction in launch vehicle costs
- Industry bus
 - * Extensive modifications required to RSDO bus (BCP-600)
 - * Custom design bus selected as baseline implementation





Acronyms

ACS	Attitude Control Subsystem
AU	Astronomical Unit
CME	Coronal Mass Ejection
COPV	Composite Pressure Vessel
DS-1	Deep Space-1 Mission
DSN	Deep Space Network
ELV	Expendable Launch Vehicle
ETR	Eastern Test Range
EUV	Extreme Ultraviolet
FSS	Far Side Sentinel
GaAs	Gallium Arsenide
HGS	High Gain System
IRU	Inertial Reference Unit





Acronyms

JPL	Jet Propulsion Laboratory
LGA	Low Gain Antenna
LWS	Living With a Star
MGA	Medium Gain Antenna
MIT	Minimum Impulse Thrusters
RSDO	Rapid Spacecraft Development Office
SEO	Sun-Earth-Orbit Centroid Angle
SEP	Sun-Earth-Platform Angle
TCM	Trajectory Control Maneuvers
TRL	Technology Readiness Level

